MORRIS

(NASA-CR-190350)
HYDROCARBON-FUELED RAMJET/SCRAMJET
TECHNOLOGY PROGRAM. PHASE 2: TEST
PLAN (United Technologies Research
Center) 67 p

N92-71123

Unclas

Z9/07 0091381

UNITED
TECHNOLOGIES
RESEARCH
CENTER

UNITED TECHNOLOGIES RESEARCH CENTER

East Hartford, Connecticut 06108

UNITED TECHNOLOGIES RESEARCH CENTER

East Hartford, Connecticut 06108

R85-956889-2

Hydrocarbon-Fueled Ramjet/Scramjet Technology Program - Phase II Test Plan

by

I. W. Kay

Contract NAS1-17794
NASA Langley Research Center

DATE July 1985

NO. OF PAGES 67

COPY NO. ______

R85-956889-2

Hydrocarbon-Fueled Ramjet/Scramjet Technology Program - Phase II Test Plan

by

I. W. Kay

United Technologies Research Center
East Hartford, Connecticut

Contract NAS1-17794
NASA Langley Research Center
July 1985

Hydrocarbon-Fueled Ramjet/Scramjet Technology Program - Phase II Test Plan

TABLE OF CONTENTS

SUMMARY

United Technologies Research Center (UTRC) has prepared a test plan for hydrocarbon ramjet/scramjet combustor testing under Phase II of a NASA LaRC program to provide combustor component technology for a research-scale scramjet engine model. This test plan addresses the development of the combustor technology applicable to the modular ramjet/scramjet engine concept devised by UTRC for successful implementation of an underslung, hydrocarbon-fueled engine for supersonic/hypersonic missiles. Execution of this test plan will provide a firm basis for the design of this engine through two major test stages: a Combustor Design Definition Stage in which critical design criteria for an axisymmetric modular combustor will be generated and a Combustor Performance Evaluation Stage during which the essential component technologies will be merged to provide a convincing demonstration of the combustor operation over the appropriate range of simulated flight condition.

The Test Plan comprises seven (7) test series of which four (4) comprise the basic program (Test Series 1 through 4) and three (3) are optional (Test Series 5 through 7). The test series, arranged in the order in which they would occur should NASA elect to sponsor the optional tests, are:

COMBUSTOR DESIGN DEFINITION STAGE (2-D)

Test Series 1 - Pilot Energy Requirement Tests

Test Series 2 - Air-breathing Pilot/Fuel Injector Performance Tests

Test Series 5 - Flame Propagation Tests (optional)

Test Series 6 - Subsonic Flameholder Tests (optional)

COMBUSTOR PERFORMANCE EVALUATION STAGE (Axisymmetric)

Test Series 3 - Supersonic Combustion Tests/High Altitude

Test Series 7 - Subsonic Combustion Tests (optional)

Test Series 4 - Supersonic Combustion Tests/Low Altitude

The test series contained within the Combustor Design Definition test stage will be conducted using two-dimensional hardware to develop design data for the final specification of an axisymmetric direct-connect combustor. Definition of pilot energy requirements and evaluation of the effectiveness of the air breathing pilot/fuel injector will be accomplished using an existing test model during Test Series 1 and 2. Optional test series to evaluate the effects of geometric variables and fuel additives on flame propagation in the supersonic combustor and to confirm the performance of a flameholder for subsonic ramjet operation (Test Series 5 and 6) would use test models fabricated under the contract. Following completion of the Combustor Design Definition test stage and finalization of the model design, an axisymmetric combustor model will be fabricated and subjected to extensive connected-pipe performance evaluation tests under the Combustor Performance Evaluation test stage. A full-scale axisymmetric combustor model will be tested in the supersonic combustion mode over a range of simulated freestream Mach numbers and altitudes during Test Series 3 and 4. Under the optional Test Series 7, the axisymmetric combustor would be tested in the subsonic combustion mode.

The performance of the optional flame propagation, subsonic flameholder and subsonic combustion tests would provide an experimental basis for defining fuel additive requirements and selecting appropriate ranges for critical scramjet/ramjet geometric parameters (such as wall contour, fuel injector locations and flameholder size and location) for the axisymmetric combustor and would enhance the fundamental understanding of the governing processes. However, should the optional tests not be conducted because of budgetary constraints, the parametric nature of the axisymmetric test series and the concentration on supersonic combustion technology would still ensure a high level of confidence that the primary program goal of achieving satisfactory combustor performance with supersonic combustion of hydrocarbon fuel would be achieved.

All of the Combustor Design Definition tests and most of the Combustor Performance Evaluation tests will be conducted in an existing connected-pipe ramjet/scramjet test facility located at the UTRC Jet Burner Test Stand. The remainder of the Combustor Performance Evaluation tests will be conducted at the General Applied Science Laboratories (GASL) where a higher airflow/pressure capability will enable simulation of lower-altitude flight conditions.

This test program makes extensive use of available equipment. For example, in addition to the use of existing test facilities, an existing UTRC Two-Dimensional Supersonic Combustion Research Model will be employed to accomplish Test Series 1 and 2. Also, the existing UTRC Mobile Laser Diagnostic Apparatus will be used to perform CARS measurements. Because of this use of available facilities and equipment, it has been possible to plan a total of over three hundred (300) tests distributed over the total program. This ensures adequate attention will be given to resolve the technological issues related to use of hydrocarbon fuels in a ramjet/scramjet combustor.

The planned duration of the test program is twelve (12) months with the draft final report documenting the test results submitted fourteen and one half (14 1/2) months after program initiation.

INTRODUCTION

The test plan described herein has been prepared in accordance with NASA Statement of Work 1-09-3740.0644-B as the final task of the first phase of a hydrocarbon-fueled ramjet/scramjet technology program being conducted by the United Technologies Research Center (UTRC) under NASA Contract NAS1-17794. During the first phase of this program, a number of analytical tasks were performed which culminated in the formulation of a conceptual design for a direct-connect supersonic flow combustor model. This test plan describes the subsequent experimental activities to be performed as the second phase of the program. A final report on the first phase of the contract describing the engine concept that this test plan addresses is presented in Ref. 1.

TEST PROGRAM

The individual experimental test series of the two major experimental stages of the test program are listed in Table 1. The optional test series are intermingled with the others to reflect the order in which the entire test program would be accomplished. The Combustor Design Definition test stage (Test Series 1, 2, and 5, 6) will be conducted using two-dimensional test models and will provide critical design criteria for the design and fabrication of the axisymmetric combustor model that will be used in the subsequent Combustor Performance Evaluation test stage (Test Series 3, 4, and 7). This program approach, in which two-dimensional test models will be used in the early test stage, was taken to allow the widest possible range of flexibility in varying the geometry of the test configuration during the design definition tests within reasonable cost and time constraints. The applicability of the two-dimensional test results to the axisymmetric configuration will be ensured by properly simulating flight conditions (i.e., Mach number, pressure and temperature) and preserving the same length scale in both configurations. Summaries of the test series objectives and descriptions of the particular tests to be performed during each of the program test series are presented below. To best reflect the program logic, the various test series are discussed in the order in which they would occur; that is, the discussions of optional test series 5, 6, and 7 are intermingled with the discussion of Test Series 1 through 4.

Combustor Design Definition Stage

Test Series 1 - Pilot Energy Requirement Tests

The objective of the pilot energy requirement tests is to define the levels of flow rate and exhaust temperature for a hot-gas generator pilot required to initiate a propagating flame in a supersonic stream. A gaseous hydrocarbon fuel, such as propane, will be used in these tests to isolate fuel vaporization considerations from the remaining processes. Using a pre-vaporized and pre-mixed fuel/air mixture will ensure that the flame propagation characteristics are not influenced by configuration-dependent fuel vaporization or mixing processes and will provide results that can be correlated against precisely-known local equivalence ratios at the pilot location. The use of an externally-fueled hot-gas generator as a pilot source for Test Series 1 will provide the desired parametric control of the pilot flow rate and exhaust temperature independently of the mainstream conditions; however, the pilot exhaust stream will be directed into the mainstream through a three-dimensional shroud that will simulate the geometry of one of the three air-breathing pilots of the axisymmetric combustor design.

A schematic diagram showing the test configuration to be employed during Test Series 1 and listing the primary test variables along with associated baseline values is presented in Fig. 1. A photograph of this existing hardware, which has been used during UTRC's 1985 Corporate-sponsored research program, is shown in Fig. 2. More detailed descriptions of the test hardware are presented in the Test Models section of this test plan. The baseline values of the combustor entrance (i.e., simulating engine station 1) Mach number (M_1), stagnation temperature (T_{T1}), and static pressure (P_1) correspond to simulated flight Mach numbers of 5.6 (at an altitude of 78,000 ft) and 7.0 (at an altitude of 104,000 ft), respectively. An existing two-dimensional facility nozzle, fabricated under the UTRC Capital Improvement program, will be used to produce the Mach 3.04 combustor entrance condition and a new nozzle, fabricated under the UTRC 1985 Capital Improvement program, will be used to produce the Mach 3.64 condition. At each simulated Mach number, a matrix of tests will be conducted in which perturbations to the combustor entrance stagnation temperature and static pressure and systematic variations in the remaining primary variables will be made. A proposed Test Series 1 test matrix is presented in Table 2. As indicated by the number of tests, the primary emphasis of the pilot energy requirement tests would be placed on the lower temperature, Mach 3.04 combustor entrance test conditions. It should also be noted that the multipoint mainstream fuel injector (Fig. 1) has been designed to minimize the fuel residence time in the duct upstream of the pilot station to preclude autoignition of the main propane at the relatively high temperatures associated with the simulated flight conditions.

Each of the piloting tests will be directed toward an assessment of whether or not combustion of the mainstream hydrocarbon fuel is initiated by the pilot. This assessment will be made on the basis of analysis of wall static pressure distributions in the test section and by visual observation of the mainstream flow in the vicinity of the pilot as afforded by sidewall windows in the two-dimensional test model. Should a wide range of required pilot mass flow rates be indicated by the Test Series 1 results, then it may be necessary to modify the pilot hardware (i.e., change the size of the pilot shroud) during these tests.

Test Series 2 - Air Breathing Pilot/Fuel Injector Performance Tests

The primary objectives of these tests are (1) to evaluate the operating characteristics (i.e., captured flow rate and exhaust temperature) of the prototype air-breathing pilot configuration, (2) to evaluate the effectiveness of both the piloting and non-piloting (i.e., downstream) fuel injectors in terms of flash-vaporizing the hydrocarbon fuel, and (3) to characterize the local fuel/air ratio distributions produced by the candidate fuel injector configurations. The Test Series 2 experimental results will be evaluated with respect to the pilot energy requirements defined in Test Series 1 to confirm the adequacy of the air-breathing pilot for stabilizing a flame in a supersonic air stream. By conducting the Test Series 2 in two-dimensional hardware early in the test program, it will be possible to implement any required modifications to the initial pilot and/or injector configurations with a greater level of flexibility than could be achieved in the axisymmetric model tests of Test Series 5 through 7. Furthermore, except for simulation of the slight curvature (circumferential) of the external wall, the wall-mounted pilot and injector geometries will be closely representative of the configurations that will be installed in the axisymmetric model.

These tests will be conducted in the same existing two-dimensional supersonic flow facility as was used in Test Series 1 with the exception that the midstream pre-mixing fuel injector assembly and the hot-gas generator pilot will be removed and will be replaced by a wall-mounted air-breathing pilot assembly (which also serves as an upstream main fuel injection source) and a candidate flash-vaporizing wall-mounted fuel injector located downstream. A selected hydrocarbon fuel such as JP-7, RJ-5 or RJ-6 will be used during these tests. Both the pilot assembly and the downstream fuel injector assembly will be installed in line with existing probe stations on the lateral centerline of the duct. The mainstream flow conditions will be set to simulate flight at Mach 5.6 and 7.0, corresponding to combustor entrance Mach numbers of 3.04 and 3.64, respectively. The corresponding baseline combustor entrance stagnation temperature and static pressure will be set as in Test Series 1.

A proposed Test Series 2 test matrix is presented in Table 3. During the initial tests, the captured flow rate of the pilot will be determined, via analysis of internal static pressure measurements in the pilot (precalibrated), for a range of mainstream flow conditions (T_{T_1} and P_1). These tests will be conducted without either pilot fuel or mainstream fuel injection and will serve to confirm the aerodynamic design of the pilot without the complication of combustion factors. Subsequent tests will then be conducted in which pilot fuel will be injected into the recirculation region of the pilot assembly and measurements will be made to determine the pilot exhaust temperature (using a double-sonic-orifice probe) and the captured pilot flow rate under burning conditions. During these pilot performance tests, it will be necessary to establish a baseline fuel flow rate through the main injection circuit of the piloting injector assembly for cooling purposes. On completion of the pilot performance tests, the Test Series 2 activities will continue with tests dedicated to the measurement of fuel temperatures in the piloting and mainstream fuel injectors (to verify the flash-vaporizing performance of those devices) and to the determination of the resulting fuel/air ratio distribution profiles in the mainstream at stations immediately downstream of the active injection site. In those tests in which the piloting injector is used to inject mainstream fuel, the pilot will be ignited to provide the proper thermal environment to the flash-vaporizing fuel passages. For those cases in which only the downstream injector is used, the pilot will be inactive thereby presenting a conservative thermal environment (i.e., without upstream combustion) for the flash-vaporization of the downstream fuel. A non-burning fuel simulant may be substituted for the mainstream hydrocarbon fuel during these tests to facilitate the measurement of the fuel/air ratio distribution profiles; the profiles will be determined through use of traversing probes and gas analysis techniques appropriate to the selected fuel or fuel simulant.

Test Series 5 - Flame Propagation Tests (optional)

The objective of the optional flame propagation tests is to evaluate (1) the flame spreading characteristics of the candidate flash-vaporizing, piloting and non-piloting, wall-mounted fuel injectors under supersonic mainstream flow conditions and (2) the effects of fuel additives on the resulting flame propagation rates. These tests would be conducted in a specially-fabricated, variable-geometry two-dimensional test installation. In this installation, the pilot and injector configurations and the axial and lateral spacings between elements will closely simulate the spatial patterns appropriate to the axisymmetric combustor model. The two-dimensional installation also will afford a convenient means for varying the divergence angle of the upper wall of the test section as a way to evaluate the anticipated strong effect of the rate of increase of combustor area ratio on flame propagation rate. In addition, the two dimensional geometry affords the flexibility to locate downstream fuel injectors at various axial and lateral positions on the lower wall. This provides an efficient means of parametrically determining the requirements for staged fuel injection to achieve high combustion efficiencies while precluding thermal choking under various simulated combustor operating conditions.

A proposed test matrix for Test Series 5 is presented in Table 4. The baseline fuel to be used during these tests will be a selected liquid hydrocarbon such as JP-7, RJ-5 or RJ-6. The effects of adding small percentages of an ignition improver, such as octyl nitrate, to the baseline fuel on the resulting ignition delay characteristics will be evaluated in the initial tests of the series. Based on those results, a fuel composition (either neat or containing a selected additive concentration) will be specified for the remainder of the test program. The subsequent portions of the Test Series 5 test matrices to be followed for each of the two supersonic combustor entrance Mach numbers, 3.04 and 3.64, will be similar. At each Mach number, the test matrix is configured to investigate independently the effects of mainstream stagnation temperature, mainstream static pressure, mainstream equivalence ratio (including various upstream/ downstream fuel flow splits), axial distance to the downstream injector and test section divergence on the resulting flame propagation characteristics. The pilot conditions will not be varied parametrically during these tests, but the pilot may be operated at different temperature levels for the different mainstream conditions if so indicated by the Test Series 1 and 2 results. It should be emphasized that the allocation of tests itemized in Table 4 to the associated test objectives is a preliminary estimate that could change significantly on the basis of trends observed in the earlier tests; however, the total number of tests allocated to the Test Series 5 effort is a true indicator of the effort that will be expended relative to the other test series of the proposed Phase II program.

The primary test measurements that will be made during Test Series 5 will comprise wall static pressure distributions in the test section, double-sonic-orifice temperature measurements at the combustor exit and visual observations of the mainstream flow as afforded by sidewall windows located in the vicinity of the pilot and the downstream fuel injectors. During selected tests, non-intrusive optical techniques also will be employed using the available windows for diagnostic purposes. These techniques will comprise either Schlieren photography or coherent anti-Stokes Raman spectroscopy (CARS).

Test Series 6 - Subsonic Flameholder Tests (optional)

The primary objective of this optional test series is to confirm the adequacy of the bluff-body flameholder design for stabilizing a flame under subsonic operating conditions (i.e., in the ramjet mode) with a selected hydrocarbon fuel such as JP-7, RJ-5 or RJ-6 (possibly containing a selected percentage of an ignition improver such as octyl nitrate. In an actual mission application, the bluff-body flameholder would be ejected from the combustor during transition from the ramjet to the scramjet mode of operation. The Test Series 6 evaluation will be conducted at a selected subsonic combustor entrance Mach number over a range of stagnation temperatures and static pressures simulating ramjet operation in the flight Mach number range from 3.5 to 5.6. The testing will be conducted in an unpiloted mode using the same basic two-dimensional hardware used in Test Series 5, but configured as shown in Fig. 3. Should this optional design confirmation test series not be selected, then a greater degree of risk would be incurred in the subsequent optional subsonic combustion tests to be conducted in the axisymmetric combustor test hardware (Test Series 7).

In the subsonic test configuration, a supersonic two-dimensional nozzle will still be located between the vitiating air heater and the two-dimensional test section. However, a downstream cooled throttle valve (in the connection to the exhaust duct) will be positioned to stabilize a shock in the constant-area portion of the combustor thereby simulating the actual upstream flow conditions and creating a subsonic flow condition at the flameholder, Station 4. Initially, a baseline flameholder, sized in accordance with available subsonic flame stability correlations for hydrocarbon fuels and having a square-shaped base region, will be located on the lateral centerline of the lower wall of the duct within the combustor and in line with an upstream fuel injector. Additional flameholders, having different characteristic dimensions (h), will be tested parametrically as part of the subsonic test series.

A proposed test matrix for Test Series 6 is presented in Table 5. The matrix shows the systematic variations that will be made in the following primary test parameters: flameholder size, combustor entrance stagnation temperature and static pressure, wall divergence angle, and local equivalence ratio. As before, the detailed allocation of tests to the specific objectives shown in Table 5 should not be considered as inflexible. Some variations to the matrix may be made on the basis of trends observed in the initial tests; however, the total number of tests allocated to Test Series 6 does reflect the intended scope of the tests relative to the other test series.

Similar instrumentation to that employed during the Test Series 5 Flame Propagation Tests will be used during Test Series 6. The measurements will be made primarily to ascertain whether or not a stabilized flame is promoted by the flameholder under the various test conditions. Efforts to optimize combustion efficiency will be reserved for the later Combustor Performance

Evaluation tests that will be conducted in the axisymmetric test hardware which is more representative of the actual engine configuration.

Combustor Performance Evaluation Stage

Test Series 3 - Supersonic Combustion Tests/High Altitude

The objective of the Test Series 3 Supersonic Combustion Tests is to evaluate the performance of a full-scale axisymmetric combustor model, representative of a geometry that would be applicable to a dual module engine configuration, operating in the supersonic mode over a range of test conditions representative of high-altitude flight conditions. The test conditions would simulate those that would prevail following ramjet to scramjet mode transition. The baseline axisymmetric combustor model design will be based heavily on the results of the design definition tests of Test Series 1 and 2 and the optional Test Series 5 and 6, but the model will also possess the flexibility to alter the geometry to accommodate anticipated interactive effects among the pilot and fuel injector components. The tests would be conducted with a representative hydrocarbon fuel such as JP-7, RJ-5 or RJ-6 (possibly containing a selected percentage of an ignition improver such as octyl nitrate).

A detailed description of a preliminary design for the axisymmetric combustor model is presented in the Test Models section of this test plan. The model will be tested utilizing two axisymmetric nozzles which are being added to the UTRC Supersonic Combustion Test Facility under the approved 1985

Capital Improvement program. These nozzles are designed to provide combustor entrance Mach numbers of 3.04 and 3.64 for the supersonic tests of the axisymmetric combustor model, simulating flight conditions of Mach 5.6 and 7.0, respectively. The facility vitiating air heater will provide the correct simulation of the combustor entrance stagnation temperature and pressure for flight at altitudes of 78,000 ft and 104,000 ft for the two respective flight Mach numbers. Evaluation of combustor performance at higher dynamic pressure levels, corresponding to flight at lower altitudes, will be accomplished during Test Series 4.

A proposed test matrix for the Test Series 3 tests is presented in Table 6. The primary test variables that will be investigated during these tests will be combustor entrance Mach number, wall contour configuration, fuel injector configuration (different combinations of locations), total mainstream equivalence ratio and upstream/downstream fuel flow split. The wall contours designated I, II and III in Table 6 refer to the three axisymmetric segment arrangements (alternative combustor geometries) described in the Test Models section of the test plan. In the event that satisfactory results are achieved with the first or second contour configuration, the test matrix may be revised to place greater emphasis on that configuration. Although the allocation of tests among the primary variables would then be shifted, the total effort of Test Series 3 will still be maintained at a level corresponding to approximately 100 tests. Also, as noted in Table 6, for a selected combustor and fuel injector configuration some of the Test Series 3 tests will be dedicated to an evaluation of the sensitivity of combustor performance to small changes in the combustor entrance stagnation temperature and static pressure levels. Although the pilot operating conditions are not listed as a primary test variable for Test Series 3, it is recognized that some variations of pilot conditions from the baseline defined on the basis of the Test Series 1 and 2 tests may be required and will be implemented as needed for satisfactory operation in the axisymmetric test hardware.

The axisymmetric test model components will be configured with an extensive array of wall static pressure taps to serve as local indicators of combustion within the test combustor. Pressure distribution measurements will be recorded during each of the Test Series 3 tests. A combustor exit instrumentation section will house a fixed rake for exhaust pitot pressure measurements, a traversable water-cooled double-sonic-orifice probe for exhaust temperature measurements, and a pair of sidewall windows for non-intrusive measurement of the combustor exhaust temperature and gas composition using coherent anti-Stokes Raman spectroscopy (CARS). Detailed combustor exit measurements will be made during selected tests for accurate assessments of overall combustion efficiency.

Test Series 7 - Subsonic Combustion Tests (optional)

The objective of the optional Test Series 7 Subsonic Combustion Tests is to evaluate the performance of the axisymmetric combustor model under subsonic operating conditions, i.e., in the ramjet mode. For these tests the combustor will be configured with the wall contour that provided the best supersonic performance during Test Series 3. The model configuration will be modified for the subsonic tests by installing a series of three identical bluff-body flameholders (evenly spaced around the periphery of the combustor) and associated downstream fuel injectors in the axisymmetric hardware as described in the Test Model section of the test plan. In a manner similar to that employed during the two-dimensional subsonic Test Series 6 tests, a cooled back-pressure valve in the axisymmetric exhaust duct will be used to maintain subsonic flow conditions in the test model during these tests.

A proposed test matrix for Test Series 7 is presented in Table 7. These tests will be conducted at a common subsonic combustor entrance Mach number and within a fixed geometry axisymmetric test section. Although a baseline flameholder characteristic dimension (h) may have been experimentally defined previously during the optional two-dimensional Test Series 6 subsonic flameholder tests, the flameholder dimension will be retained as one of the primary test variables because of the possibility of interactive effects in the axisymmetric hardware. The other controlled parameters will comprise the distance that the flash-vaporizing fuel injector is located upstream of the flameholder, the mainstream equivalence ratio and the combustor entrance stagnation temperature and static pressure. The hydrocarbon fuel to be used for these tests will be the same as that used in Test Series 3. As denoted by the relatively small total number of tests allocated to the Test Series 7 effort, it is anticipated that significantly less developmental effort will be required to demonstrate satisfactory subsonic combustor performance in comparison to that required for the supersonic mode of operation.

The same instrumentation provisions as were used in the Test Series 3 supersonic combustor evaluation tests will also be used for the subsonic tests. Careful examination of the three-dimensional array of wall static pressure distributions in the region downstream of the flameholder will serve as a primary means of describing the ensuing combustion process. Combustion efficiency evaluations will be based on exit station temperature measurements as afforded by either double-sonic-orifice probe measurements or optical diagnostic techniques.

Test Series 4 - Supersonic Combustion Tests/Low Altitude

The objective of the Test Series 4 tests is to evaluate the supersonic combustion performance of the axisymmetric combustor model, selected on the basis of the simulated high-altitude test results of Test Series 3, under higher dynamic pressure conditions corresponding to simulated supersonic flight at lower altitudes. These final performance documentation tests will be conducted in the General Applied Science Laboratories (GASL) facilities wherein the maximum air supply pressure is 1000 psia compared to a maximum of 400 psia at the UTRC ramjet/scramjet test facility; the higher supply pressure at the GASL facility will provide a capability to conduct tests in the same combustor hardware and with the same tunnel nozzles that were used in Test Series 3 at static pressure levels that will be higher by a factor of approximately 2.5.

A proposed test matrix for the Test Series 4 tests is presented in Table 8. The initial test series at GASL will be conducted at test conditions identical to those previously established during the UTRC tests (during Test Series 3), and with the same hydrocarbon fuel and at the same levels of mainstream equivalence ratio and fuel injection staging ratio (i.e., upstream-to-downstream fuel flow splits) for which comparable performance data will be available. The results of these tests will serve to validate the equivalency of results from the two facilities. The remaining GASL tests will then be conducted with the same fuel at combustor entrance conditions simulating flight at Mach 5.6 and Mach 7.0 at the lowest altitudes compatible with the test facility. The corresponding combustor entrance conditions for the Mach 5.6 flight condition are M_1 = 3.04, T_{T_1} = 2675R and P_1 = 25 psia (53,000-ft altitude); for the Mach 7.0 flight condition, the corresponding values are M_1 = 3.64, T_{T_1} = 3788R and P_1 = 10 psia (78,000-ft altitude). Parametric variations in the fuel injector configuration, the mainstream equivalence ratio and the upstream/downstream fuel flow split will be performed for each simulated flight condition. The Test Series 4 test matrix is presented on the basis that a combustor with a fixed wall contour will provide satisfactory supersonic combustion performance at the higher pressure levels associated with these tests; however, should the early test results show that thermal choking becomes a problem that cannot be accommodated by fuel injector location and fuel staging variations alone, then it may be necessary to modify the model configuration (using available combustor segments from the Test Series 3 activity) during the Test Series 4 effort. In any event, the total number of tests allocated to the Test Series 4 will remain as shown in Table 8. The instrumentation to be used for the Test Series 4 tests will be identical to that used during the supersonic combustion tests of Test Series 3, as described previously.

TEST MODELS

The combustor models to be tested under the Phase II program will be of boilerplate, heat-sink construction and will include three separate installations to accommodate the needs of the seven (four without options) separate experimental tasks. The first two test series of the Combustor Design Definition stage will be conducted in an existing two-dimensional test installation while the optional test series (5 and 6) will be conducted in a new, specially-fabricated two-dimensional installation having a wide variety of variable-geometry features. The final three test series of the Combustor Performance Evaluation stage will be conducted in axisymmetric hardware which provides a full-scale simulation of the actual combustor geometry of the anticipated hydrocarbon ramjet/scramjet engine.

Existing Two-Dimensional Research Model

A photograph of the existing two-dimensional supersonic combustion research model installed in the UTRC Ramjet/Scramjet Test Facility is presented in Fig. 2. The test section geometry is shown in Fig. 4. This model will be used in the first two test series of the Phase II program.

The test section entrance is 3-in-high x 6-in-wide and connects directly to the exit station of the two-dimensional tunnel nozzle. The test section cross-sectional area remains constant for the first six inches. The upper wall then diverges at an angle of one-half degree for the next eighteen inches and at an angle of three degrees for the remaining length. The overall length from the entrance station to the exit, where the flow is dumped into a circular cross-section exhaust pipe, is 84 inches.

The two-dimensional hardware includes provisions for mounting a variety of pilot configurations on the lateral centerline of the lower wall near the combustor entrance. For Test Series 2, the pilot configuration will be as shown in Fig. 5 wherein the pilot sub-assembly also serves to inject flash-vaporized fuel into the mainstream flow from its external (cowl) surface. Other provisions of the test hardware that will facilitate the conduct of Test Series 2 include the capability for mounting a non-piloting, flash-vaporizing fuel injector on the lateral centerline of the lower wall at one of a series of possible locations downstream of the pilot site. A schematic representation of the configuration of a non-piloting flash-vaporizing fuel injector is presented in Fig. 6. When configured for the initial pilot energy requirement test series (Test Series 1), the pilot configuration will comprise a shroud (simulating the exit geometry of the pilot of Fig. 5) through which the flow from an externally-mounted hot-gas generator will be directed into the mainstream. During those tests a pre-mixing fuel injector will be located upstream of the tunnel nozzle throat (cf., Fig. 1) in the vicinity of the large circular flange shown in the photograph of Fig. 2. The pre-mixing fuel injector comprises an array of five streamlined, vertically-oriented struts housing eight injection sites each with the resulting forty injection sites distributed uniformly across the nozzle cross-section at a subsonic flow station.

The existing two-dimensional hardware incorporates two pairs of opposed viewing ports in the sidewalls (as shown in Fig. 2) in the vicinity of the piloting and non-piloting fuel injectors for flow visualization or optical measurement purposes. In addition, locations for traversing probes are provided just downstream of both the piloting and non-piloting fuel injectors which can be used for either direct temperature measurements (using double-sonic-orifice probes) or gas composition measurements. Other instrumentation provisions in the existing two-dimensional assembly include forty-eight static pressure taps, distributed both laterally and axially on the four walls of the test section.

Variable-Geometry Two-Dimensional Test Model

A drawing of the basic two-dimensional assembly that will be used in the optional Test Series 5 of the Phase II program is presented in Fig. 7. The test section entrance will be 3-in-high x 6-in-wide and the overall combustor length (including the section housing the pilot) will be 58 in. Provisions to vary the combustor contour will include height adjustment mechanisms at two intermediate stations within the test section and an adjustable flange at the combustor exit. For a given test configuration, the top and bottom walls of the combustor will be clamped between the flat sidewalls of the test assembly and the assembly will be made leak-tight using high-temperature silicone rubber seals. A summary of the baseline dimensions of the variable-geometry two-dimensional test hardware is presented in Table 9.

The two-dimensional hardware will include provisions for mounting a piloting, flash-vaporizing fuel injector on the lateral centerline of the lower wall near the combustor entrance and for locating a pair of laterally spaced flash-vaporizing fuel injectors at one of three possible stations downstream of the pilot. Drawings showing the details of preliminary configurations of the piloting and non-piloting fuel injectors are presented in Figs. 5 and 6, respectively. As shown in Fig. 7, the two-dimensional hardware will incorporate three pairs of quartz sidewall-mounted windows in the vicinity of the piloting and non-piloting fuel injectors for flow visualization or optical measurement purposes. The rectangular flanged window design indicated in Fig. 7 is one that has been shown to be effective under similar scramjet operating conditions in previous tests at UTRC (Refs. 2, 3). In addition to the test windows, locations for traversing DSO probes also will be provided just downstream of both the piloting and non-piloting injectors for direct temperature or gas composition measurements. Other instrumentation provisions in the two-dimensional assembly will consist of an array of approximately 100 wall static pressure taps, distributed both laterally and axially on the flat lower wall and an additional 20 to 30 wall static pressure taps distributed on the other three walls of the test section.

Should the optional subsonic flameholder Test Series 6 be selected, the two-dimensional assembly also will provide the capability for mounting a bluff-body flameholder on the lateral centerline of the lower wall at an axial station downstream of the intermediate height adjustment mechanism and for locating an associated flash-vaporizing fuel injector (identical to the configuration shown in Fig. 6) in line with and slightly upstream of the flameholder. An additional pair of sidewall-mounted quartz windows, shown in Fig. 7 near the combustor exit station, will provide optical access to the combustor flow field at a location appropriate to the subsonic combustor tests.

Axisymmetric Test Model

A drawing of the axisymmetric test hardware that will be used during the Combustor Performance Evaluation stage of the Phase II program is presented in Fig. 8. These combustor test sections are full-scale models of the combustor module which would be used in the actual propulsion system. Three separate configurations are shown, each having a combustor entrance duct radius of 2.05 in., an overall length of 58 in., and an overall combustor area ratio (i.e., exit area/entrance area) of 2.5. The three configurations differ primarily in the length of the upstream portion of the test section over which the cross-sectional area is held essentially constant; with the combustor exit area held constant, the resulting divergent half-angles of the downstream portions of the test section vary in the range from 1.7 to 2.8 deg. A summary of the geometric dimensions of the three axisymmetric combustor models is presented in Table 10. It should be noted that the overall geometry (i.e., length and area ratio) of the axisymmetric models is identical to that of the baseline two-dimensional hardware used in the earlier Design Definition Stage of the test program. Because the upstream and downstream axisymmetric combustor segments all mate with a common flange dimension, the model hardware also provides a capability to vary the overall combustor length by interchanging the various sections.

A series of three piloting, flash-vaporizing fuel injectors are evenly spaced around the periphery of the combustor model at the entrance station. Provisions for locating a supplementary set of three non-piloting, flash-vaporizing fuel injectors at an axial station 4, 6 or 8 inches downstream of, and interdigitated between the pilots are included in each of the upstream constant-area combustor sections. A similar set of three peripherally-spaced flash-vaporizing fuel injectors may also be located at one of three available mounting locations at the upstream end of each of the diverging downstream combustor sections. By rotating the flanged sections, the injectors in the downstream duct may be aligned axially with either the piloting or the non-piloting upstream fuel injectors. Except for the curvature required to adapt the pilots and the fuel injectors to the axisymmetric combustor cross-section, the configurations of those components will be identical to the comparable two-dimensional piloting and non-piloting flash-vaporizing fuel injectors used in the two-dimensional tests and shown in Figs. 5 and 6, respectively.

As shown in the upper centerline of Fig. 8, one of the axisymmetric combustor assemblies will include provisions to mount a flameholder assembly (comprising three peripherally-spaced bluff bodies) in the downstream diverging combustor section for use in the optional subsonic combustion tests of Test Series 7. Each flameholder will be a pyramidal-shaped bluff body having a square downstream-facing base region. The flameholder mounting ring will allow placement of each of the flameholders directly downstream of a flash-vaporizing fuel injector located in the divergent section. When configured for a subsonic test, a water-cooled butterfly valve located in the connection to the exhaust duct will be used to control the combustor entrance Mach number.

An axisymmetric combustor exit instrumentation section will be located downstream of the divergent section of each combustor for the determination of combustor performance. The instrumentation section will house a water-cooled pitot pressure rake, a traversable water-cooled double-sonic-orifice probe and a pair of diametrically-opposed quartz windows for optical measurement purposes. Other instrumentation provisions in the axisymmetric combustors will comprise arrays of wall static pressure taps distributed axially and peripherally in the test sections. The total number of taps in any one assembly will be approximately 100. Wall static pressure taps also will be included in a short constant area axisymmetric spool piece located between the facility nozzle and the combustor entrance station to facilitate detection of an unstart condition (e.g., as might be caused by inlet-combustor interaction or thermal choking) in the test assembly.

TEST FACILITIES

UTRC Ramjet/Scramjet Test Facility

The majority of the proposed Phase II test program will be conducted in a connected-pipe type ramjet/scramjet test facility located at the Jet Burner Test Stand (JBTS) of UTRC. The JBTS is a self-contained combustion facility, having seven test cells. The air system provides continuous airflows of up to 10 lb/sec at pressures up to 400 psia; higher airflow rates of up to 150 lb/sec may be obtained in a transient mode using three 5000 ft^3 storage tanks. Air heating is accomplished in a gaseous hydrogen-fueled, water-cooled heater assembly. Oxygen is injected into the upstream portion of the heater to replenish the oxygen consumed in the hydrogen combustion process and to obtain an oxygen mole fraction in the exhaust gas equal to that of air. The present heater is being replaced under an approved 1985 Capital Improvement Project to extend the range of operating conditions to gas temperatures up to 4500 R at pressures up to 400 psia. The design of this new vitiating heater is similar to that of an existing unit presently being used at the United Technologies Chemical Systems Division.

A drawing showing the installation of either a two-dimensional or axisymmetric combustor model in the test facility is presented in Fig. 9. Depending upon the cross-sectional shape of the combustor model, the heated airstream is directed through either a two-dimensional or axisymmetric water-cooled nozzle assembly to accelerate the flow to the desired combustor entrance Mach number condition. These nozzles are part of the UTRC Ramjet/Scramjet Test Facility. A two-dimensional, nozzle designed to deliver a Mach 3.04 combustor entrance flow (simulating Mach 5.6 flight) to a 3-in. x 6-in. combustor entrance cross-section compatible with the two-dimensional models to be tested during the Phase II program is presently available. An additional two-dimensional Mach 3.64 nozzle (simulating Mach 7.0 flight) and axisymmetric Mach 3.04 and Mach 3.64 nozzles will be fabricated under UTRC's 1985 Capital Improvement program and will be available for use in the test program. The axisymmetric nozzles will be compatible with the 4.1-in. combustor entrance diameter of the axisymmetric combustor models. They also will be used for the higher-pressure model tests to be conducted later in the test program at the GASL facility.

After passing through a combustor-exit instrumentation section, the exhaust flow from the combustor model is cooled with quench water and directed through an air-driven exhaust ejector to the atmosphere. The ejector operates at an air flow rate of approximately 125 lb/sec and is capable of maintaining separation-free conditions in a supersonic-flow combustor at static pressure levels as low as 2 psia.

A wide variety of auxiliary systems for handling liquid fuels, gases and cooling water are available at the JBTS. Hydrocarbon fuels are supplied to the test cells at pressures of up to 1500 psia and at flow rates up to 37.5 gpm. Hydrogen, oxygen and nitrogen gas systems, at pressures up to 2400 psi, also are provided. Cooling water is provided from either a city-water system at flow rates up to approximately 1000 gpm or a high-pressure (300 psi) pump-fed system at flow rates up to 100 gpm.

An existing JBTS data acquisition system will be employed for the Phase II tests. This system consists of a high speed (20 channel) analog-to-digital converting system. It is designed to accept outputs from 48-port pressure Scanivalves, 26-junction temperature scanners, single pressure transducers and turbine flowmeters. The total time required for a complete data scan is less than 5 sec. The digitized binary equivalent of the analog inputs is stored on Univac 1100/81A-compatible magnetic tape. The high speed system uses a direct link to the Univac 1100/81A for on-line processing of the data. The reduced data are printed out at the JBTS on a DCT-500 remote terminal. A similar system can be employed to transmit selected Phase II data in a coded form by telephone to NASA Langley. Dynamic data also can be acquired and recorded on Visicorders or on a high speed FM tape unit at 20 KHz. In this case digitizing takes place off-line where the recorded data are played back at a reduced speed.

GASL Test Facility

GASL's test facilities include a blowdown type supersonic wind tunnel, two direct-connect combustor test facilities, a supersonic test facility, several atmospheric burner test stands, a machine shop, an instrumentation shop, a computer-based data acquisition system, engineering offices and mechanical design and drafting facilities. The direct-connect facilities are designed for gas turbine and ramjet combustion testing and are capable of supplying dry heated air at temperatures of 2000 R and pressures to 1500 psia. The supersonic facility is designed for ramjet and scramjet testing and supplies vitiated air to 4000 R and 1000 psia.

The basic elements of the test facility are a high-pressure air storage tank farm, a 40 ft. diameter vacuum sphere, two pebble-bed storage heaters, a hydrogen combustion driven vitiated air heater, two Norwalk and two Chicago Pneumatic air compressors, and two Beech-Russ rotary vacuum pumps. The air supply system consists of four high pressure compressors and 3500 cubic feet of 2000 psia storage tanks. When fully charged, the system holds 36,000 lbs of air. The air storage tanks can be fully recharged each 24 hours. The amount of air available for a test depends on the pressure required and the flow rate (which determines the pressure drop through the regulating system). For testing at 1000 psia, a flow rate of 10 lb/sec can be maintained for 30 minutes while a flow rate of 40 lb/sec can be maintained for approximately 5 minutes. Decreasing the test pressure level to 300 psia increases running times to 50 minutes and 9 minutes for the same 10 lb/sec and 40 lb/sec flow rates.

GASL's vitiated air heater employs in-stream hydrogen/oxygen combustion to preheat air to a maximum temperature of 4000 R. The relative amounts of hydrogen and oxygen added to the airstream are adjusted to produce a test gas with an oxygen volume fraction of 21%. The vitiated air heater employs a regeneratively-cooled refractory-lined combustion chamber with a 1200 psia pressure capability. The maximum airflow rate to the vitiated air heater is 40 lb/sec. The flows of air, hydrogen and oxygen are controlled by venturis and pressure regulators and can be programmed for either constant, ramped or stepped mass flow and/or outlet temperatures.

Auxiliary systems for handling gases, liquids and slurries (e.g., hydrogen, methane, propane, ethylene, Jet-A, JP-fuels, fuel oils, coal-water mixtures, etc.) are also in place. Instrumentation includes up to 250 channels of pressure, up to 120 channels of temperature, force and acceleration measurements, gas sampling (CO_2, CO, O_2, UHC, NO_x), and both photographic and video remote viewing systems.

Data acquisition is supported by a variety of analog devices, e.g., strip chart recorders and gages, and by two major analog-to-digital subsystems; an HP 2240 Measurement and Control Processor and a Pressure Systems Inc., electronically scanned pressure measurement system. The A/D subsystems are interfaced to GASL's HP 1000 computer system which operates in a dedicated, real-time mode during tests. The computer system provides selected data reduction on-line and complete data reduction and graphics off-line within minutes after completion of a test. It can also transmit the reduced data to the sponsor's computer within minutes after a test to permit direct interaction without requiring the sponsor to place an engineer on-site.

DATA REDUCTION AND SPECIALIZED INSTRUMENTATION

Because of the heat-sink type construction of the test hardware, all of the combustion tests to be conducted as part of the Phase II program will be limited to durations on the order of twenty seconds. Since the acquisition of a complete data recording will take approximately five seconds and some additional time will always be required for the test conditions to stabilize, each test will be dedicated to the documentation of performance for a single operating condition.

Steady-state air, replenishment oxygen and any gaseous pilot reactant flow rates will be determined using calibrated choked venturis in the respective supply lines. Liquid fuel flow rates will be determined using turbine-type flowmeters. For the bulk of the tests conducted during this program, a data reduction procedure based upon analysis of the measured wall static pressure distributions in the combustor will be used as a primary means of evaluating combustor performance. Accurate and rapid measurements of the wall static pressure distributions in the test section will be made by means of high speed pressure scanners. Quantitative results will be obtained from these pressure distributions by calculating normalized pressure-distance integrals over selected segments of the combustor (from axial station x_1 to x_2), in accordance with the expression shown below:

$$\mathcal{J} = \frac{\left[\int_{x_1}^{x_2} \frac{P_s}{P_{T_1}}\, dx\right]_{\phi} - \left[\int_{x_1}^{x_2} \frac{P_s}{P_{T_1}}\, dx\right]_{\phi=0}}{\left[\int_{x_1}^{x_2} \frac{P_s}{P_{T_1}}\, dx\right]_{\phi=0}} \qquad (1)$$

It should be noted that such pressure-distance integrals have previously been used as a means of evaluating whether or not successful flame stabilization was achieved in similar supersonic combustion experiments (Refs. 2, 3). However, since the value of the pressure-distance integral is a strong function of both the heat release rate and the actual combustor area distribution, its magnitude is not a unique measure of combustor performance. Nevertheless, because it does provide a relative measure of performance for similar geometries and is a convenient means of summarizing the test results, pressure-distance integrals will be calculated and presented for all of the test data.

In addition to providing a gross measure of combustor performance based on measured pressure distributions, the temporal variations in individual wall static pressures can be used to indicate both the onset of combustion and the extent of the region affected by that combustion. The procedures developed in Ref. 4 under the direction of Ferri will be used to estimate the combustion activity at individual fuel injection sites and to assess the performance of stages of fuel injection. These procedures are based on wave-diagram calculations that quantify the influence of combustion on the experimentally determined model static pressure variations. The wall static pressure instrumentation in the two-dimensional and axisymmetric models will be structured to enable those diagnostic procedures to be implemented.

Double-sonic-orifice (DSO) probes will be used for more quantitative assessments of combustor performance. A schematic representation of one of these probes, which is water-cooled and capable of operation in streams with stagnation temperatures as high as 5500R (i.e., above the level for which conventional thermocouples can be used) is presented in Fig. 10. With a DSO probe, the local stagnation temperature at the probe tip is calculated using the continuity equation for choked flow. The measured quantities are the pressure upstream of the choked tip orifice, and the temperature and pressure upstream of a second choked orifice located downstream of a cooling passage. A complete derivation of equations used when reducing the DSO probe data is presented in Appendix I of Ref. 2. The design of the particular DSO probes proposed for use during this program (as shown in Fig. 10) provides for minimizing the effects of thermal expansion of the probe tip orifice by locating the tip cooling water passages within approximately 0.020 in. of the throat and also includes an internal steam jacket around the main probe gas passage to ensure that gaseous flow conditions prevail between the two orifices. An illustration of the accuracy achieved with the DSO probes is presented in Fig. 11 which shows the results of calibration tests in which temperatures of a hot nitrogen stream measured with a chromel-alumel thermocouple agreed with the DSO derived temperatures within ±20R over the range from 500R to 1300R.

In addition to the DSO measurements described above, efforts will be made in selected tests to determine local static temperatures and species concentrations using a non-intrusive optical technique, coherent anti-Stokes Raman spectroscopy (CARS). The feasibility of applying CARS to supersonic combustion experiments and to using CARS for measurements in high pressure/high temperature environments (such as in a gas turbine engine) has been demonstrated at UTRC (Refs. 5 and 6). Furthermore, a mobile CARS measurement apparatus is available at UTRC that can readily be installed in the Ramjet/Scramjet test facility. This apparatus has been used recently to successfully measure temperatures in the supersonic combustor section of the UTRC Ramjet/Scramjet Test facility. A comprehensive description of the applications of CARS to combustion diagnostics is presented in Ref. 7.

Appropriate test results will be compared and correlated with analytical predictions made using the UTRC Mixing and Combustion Computer Program. This procedure (cf., Appendix G in Ref. 1) was helpful in providing a detailed evaluation of the rate and extent of combustion present under different flow condition for the selected engine configurations in Ref. 1. It will be employed in Phase II to model selected tests and provide a correlation with test results. If a mutual agreement between test and analysis could be obtained under different simulated flow conditions, the analytical procedure could then be employed with greater confidence in flow regimes beyond those addressed in the Phase II tests (e.g., higher pressure, lower altitude conditions).

TEST SCHEDULE

A test schedule for the proposed Phase II program is presented in Fig. 12. As indicated in Fig. 12, the final design and fabrication of the axisymmetric hardware that will be used in the combustor performance evaluation tests will be scheduled to enable incorporation of the results of the preliminary combustor design definition tasks. The relative durations of the separate test activities generally reflect the number of tests associated with each test series as described in Tables 2-8. The total facility occupancy time (including model installation time) will be six months if the optional test series are not selected. If the optional test series are included, the facility occupancy time would be extended to nine and one-half months. In either case, the experimental activities will be completed within twelve months of the program start. A draft final report, incorporating complete discussions of the Phase II test results, will be submitted within fifteen months of the program start.

REFERENCES

1. Karanian, A. J., et. al.: Hydrocarbon-Fueled Ramjet/Scramjet Technology Program - Phase I, Final Report. UTRC Report R85-956889-1, July 1985.

2. Kay, I. W., J. B. McVey, C. E. Kepler and L. Chiapetta: Hydrocarbon-Fueled Scramjet. Technical Report AFAPL-TR-68-146, Vol. VIII - Piloting and Flame Propagation, May 1971.

3. Kay, I. W. and J. B. McVey: Hydrocarbon-Fueled Scramjet. Technical Report AFAPL-TR-68-146, Vol. XI - Combustor Demonstration Program, May 1972.

4. Peschke, W. and E. Reiss: Diffusion Controlled Supersonic Combustion, Analysis and Design of Combostors, GASL Technical Report 672, Nov. 1967.

5. Shirley, J. A., R. J. Hall and A. C. Eckbreth: Investigation of the Feasibility of CARS Measurements in Scramjet Combustion. NASA Contractor Report 159280, July 1980.

6. Stufflebeam, J. H., J. F. Verdieck and R. J. Hall: CARS Diagnostics of High Pressure and Temperature Gases. AIAA Paper No. 83-1478, presented at the AIAA 18th Thermophysics Conference, Montreal, Canada, June 1983.

7. Hall, R. J. and A. C. Eckbreth: Coherent Anti-Stokes Raman Spectroscopy (CARS): Application to Combustion Diagnostics to appear in Laser Applications (Volume V), R. K. Erf, ed., Academic Press.

NOMENCLATURE

A	Cross-sectional area of combustor
A_1	Combustor entrance cross-sectional area
f/a	Fuel-to-air ratio
h	Characteristic dimension of flameholder
K	DSO probe calibration constant
M_1	Combustor entrance Mach number
P_1	Combustor entrance static pressure
P_S	Combustor wall static pressure
T_{T_1}	Combustor entrance stagnation temperature
$\dot{W}$	Mass flow rate
x	Axial distance
θ	Divergence angle of combustor wall
ϕ	Equivalence ratio
$\mathcal{J}$	Pressure change integral

<u>Engine Stations</u>

1	Inlet cowl lip
2	Inlet throat
3	End constant-area section
4	Flameholder location
5	Combustor exit

<u>Subscripts</u>

p	Pilot
f1	1st fuel stage (pilot external fuel)
f2	2nd fuel stage

TABLE 1. PROGRAM TEST SERIES

(Note: Test Series shown in order of occurrence)

COMBUSTOR DESIGN DEFINITION STAGE (2-D)

Test Series 1 - Pilot Energy Requirement Tests

Test Series 2 - Air-breathing Pilot/Fuel Injector Performance Tests

Test Series 5 - Flame Propagation Tests (Optional)

Test Series 6 - Subsonic Flameholder Tests (Optional)

COMBUSTOR PERFORMANCE EVALUATION STAGE (AXISYMMETRIC)

Test Series 3 - Supersonic Combustion Tests/High Altitude

Test Series 7 - Subsonic Combustion Tests (Optional)

Test Series 4 - Supersonic Combustion Tests/Low Altitude

TABLE 2. TEST SERIES 1 - PILOT ENERGY REQUIREMENT TEST MATRIX

M_2	$\dot{W}_P$ & T_{T_P}	ϕ_1	T_{T_1} & P_1	Number of Tests
3.04	Baseline + 6 Combinations	Baseline	Baseline	7
↓	2 Combinations	2 Values	Baseline	4
↓	2 Combinations	Baseline	4 Combinations	8
3.64	Baseline + 3 Combinations	Baseline	Baseline	4
↓	2 Combinations	4 Values	Baseline	4
↓	2 Combinations	Baseline	2 Combinations	4

Total number of tests = 39

Baseline Values

M_1	$\dot{W}_1$(lb/sec)	$\dot{W}_p$(lb/sec)	T_{T_P}(R)	ϕ_1	T_{T_1}(R)	P_1(psia)
3.04	12.6	0.2	5000	1.0	2675	7.7
3.64	6.0	0.1	5000	1.0	3788	3.2

TABLE 3. TEST SERIES 2 - PILOT/FUEL INJECTOR PERFORMANCE TEST MATRIX

M_1	T_{T_1} & P_1	ϕ_p	ϕ_{f1}	ϕ_{f2}	Key Measurements	Number of Tests
3.04	Baseline + 3 Combinations	None	None	None	• Captured pilot flow rate (cold)	4
↓	Baseline + 1 Combination	3 Values	Baseline	None	• Captured pilot flow rate (hot) • pilot temperature	6
↓	Baseline	Selected	3 Values	None	• heating of fuel from piloting injector • f/a profiles at upstream probe location	3
↓	Baseline	None	None	3 Values	• heating of fuel from downstream injector • f/a profiles at down-stream probe location	3
3.64	Repeat Matrix of Mach 3.04 Tests					16

Total Number of tests = 32

Baseline Values

M_1	$\dot{W}_1$(lb/sec)	T_{T_1}(R)	P_1(psia)	ϕ_{f1}
3.04	12.6	2675	7.7	1.0
3.64	6.0	3788	3.2	1.0

TABLE 4. TEST SERIES 5 - FLAME PROPAGATION TEST MATRIX (OPTIONAL)

M_2	T_{T_1} & P_2	Fuel Additive %	Mainstream Equivalence Ratio Upstream	Mainstream Equivalence Ratio Downstream	Distance to Downstream Injector (in)	Upper Wall Divergence Angle	Number of Tests
3.04	Baseline	0,1,2	5 Values	0	N/A	Baseline	10
↓	Baseline	Selected	Selected	5 Values	4, 6, 8	Baseline	12
↓	3 Combina-ations	Selected	3 Values	0	N/A	Baseline	9
↓	Baseline	Selected	3 Values	0	N/A	2 Variations	6
3.64	Baseline	Selected	5 Values	0	N/A	Baseline	5
↓	Baseline	Selected	Selected	5 Values	4,6,8	Baseline	12
↓	3 Combina-tions	Selected	3 Values	0	N/A	Baseline	9
↓	Baseline	Selected	3 Values	0	N/A	2 Variations	6

Total number of tests = 69

Baseline Values

M_1	$\dot{W}_1$ (lb/sec)	T_{T_1} (R)	P_1 (psia)	Upper Wall Divergence Angle (deg) Upstream	Upper Wall Divergence Angle (deg) Downstream
3.04	12.6	2675	7.7	0.6	8.0
3.64	6.0	3788	3.2	0.6	8.0

TABLE 5. TEST SERIES 6 - SUBSONIC FLAMEHOLDER TEST MATRIX (OPTIONAL)

Flameholder size	T_{T_1} (R)	P_1 (psia)	Θ (deg)	ϕ	Number of Tests
h	1350-2675	20-40	0.5	3 values	10
h	1350-2675	20-40	2.0	3 values	6
h	1350	Selected	4.0	3 values	3
2 additional values	1350	Selected	0.5	3 values	6

Total number of test = 25

TABLE 6. TEST SERIES 3 - SUPERSONIC COMBUSTION TEST MATRIX (UTRC)

M_2	T_{T_1} (R)	P_1 (psia)	Wall Contour	Fuel Inj. Config.	Mainstream Equiv. Ratio	Up/Down Fuel Flow Split	Number of Tests
3.04	2675	7.7	I	A	3 Values	3 Values	5
			I	B	3 Values	3 Values	5
			I	C	3 Values	3 Values	5
			II	A, B, C	3 Values	3 Values	15
			III	A, B, C	3 Values	3 Values	15
	5 Combinations		Selected	Selected	Selected	Selected	5
3.64	3788	3.2	-- Repeat Matrix of Mach 3.04 Tests --				50

Total Number of Tests = 100

TABLE 7. TEST SERIES 7 - SUBSONIC COMBUSTION TEST MATRIX (OPTIONAL)

Flameholder Dimension	Fuel Inj. Config.	Mainstream Equiv. Ratio	T_{T_1} (R)	P_1 (psia)	Number of Tests
h	Baseline	3 values	1350	20-40	5
h	Baseline	3 values	2675	20-40	5
h	2 additional values	3 values	1350	selected	6
h	selected	3 values	2675	selected	3
2 additional values	selected	3 values	1350	selected	3

Total Number of Tests = 22

TABLE 8. TEST SERIES 4 - SUPERSONIC COMBUSTION TEST MATRIX (GASL)

M_2	T_{T_1} (R)	P_1 (psia)	Fuel Inj. Config.	Mainstream Equiv. Ratio	Up/Down Fuel Flow Split	Number of Tests
3.04	2675	7.7	selected	3 values	3 values	5
3.64	3788	3.2	selected	3 values	3 values	5
3.04	2675	~ 25	2 arrays	3 values	3 values	10
3.64	3788	~ 10	2 arrays	3 values	3 values	10

Total number of tests = 30

TABLE 9. BASELINE GEOMETRY OF THE VARIABLE-GEOMETRY TWO-DIMENSIONAL COMBUSTOR

Station (Engine Station)	Axial Distance (in.)	Duct Height (in.)	Combustor Area Ratio A/A_1	Divergence Angle* (deg)
Combustor Entrance (1)	0	3.00	1.00	-3.6
Pilot Exit (2)	4	2.75	0.92	0.6
Intermediate (3)	26	3.00	1.00	8.0
Combustor Exit (5)	58	7.50	2.50	

*Between stations

TABLE 10. GEOMETRY OF AXISYMMETRIC COMBUSTOR MODELS

Station (Engine Station)	Axial Distance (in.)	Duct Radius (in.)	Combustor Area Ratio, A/A_1	Divergence* Half-Angle (deg)
Combustor Entrance (1)	0	2.05	1.00	
				-1.3
Pilot Exit (2)	4	1.96	0.42	
				0.36, 0.23, 0.16
Intermediate (3)	14, 22, 30	2.05	1.00	
				1.7, 2.1, 2.8
Combustor Exit (5)	58	6.48	2.50	

*Between stations

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

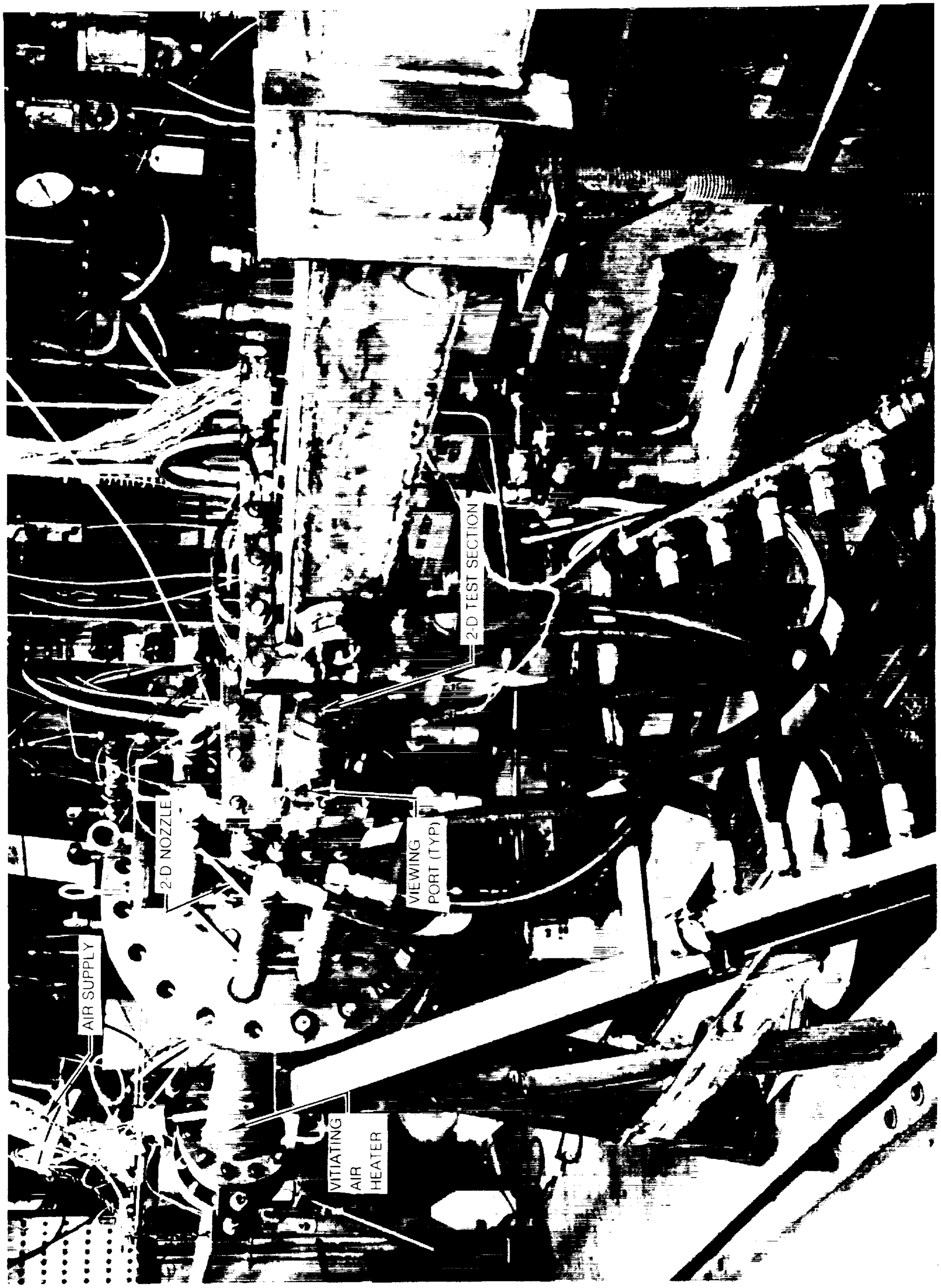
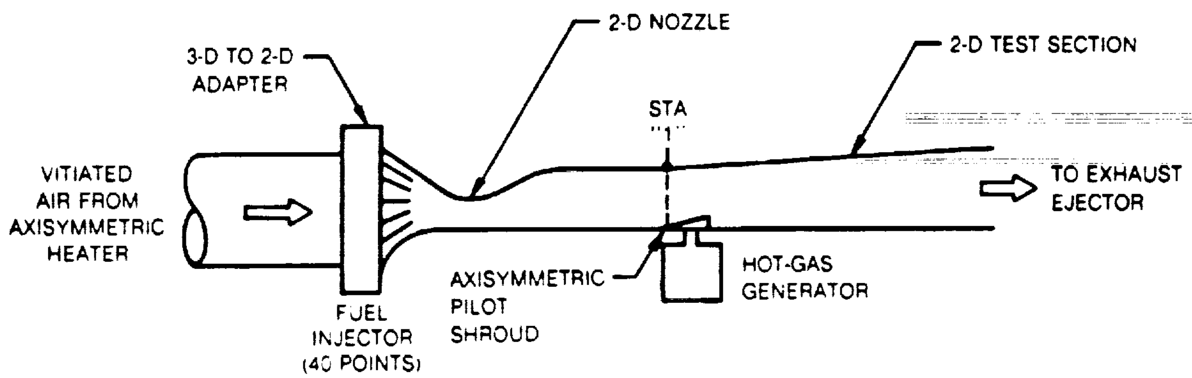


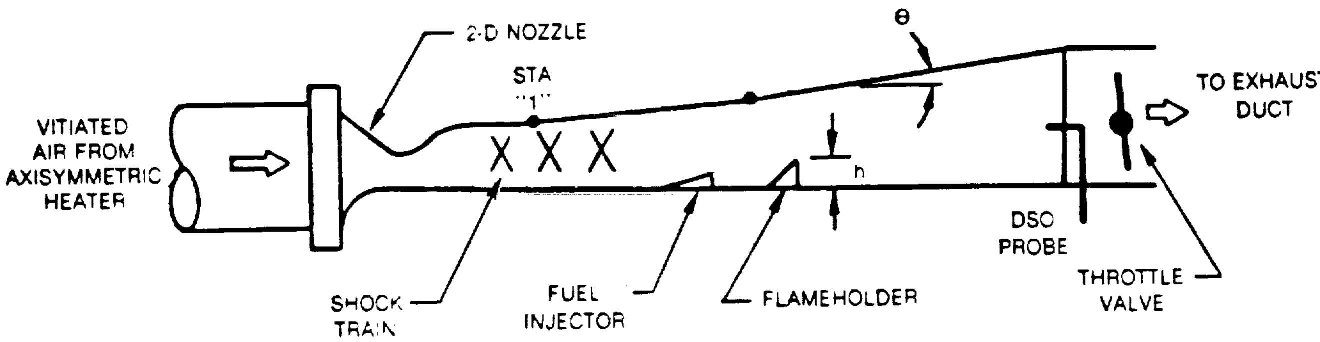
Figure 2. UTRC Ramjet/Scramjet Test Facility

85–6–17–1



PRIMARY TEST VARIABLE	BASELINE VALUE
M_1	3.04; 3.64
T_{T_1}	2675 R; 3788 R
P_1	7.7 psia; 3.2 psia
$(f/a)_1$	STOICHIOMETRIC
$\dot{W}_p$	1.7% OF $\dot{W}_1$
T_{T_p}	5000 R

Figure 1. Piloting Test Configuration



PRIMARY TEST VARIABLE	RANGE
T_{T_1}	1350 TO 2675 R
P_1	20 TO 40 psia
θ	0.5 TO 8.0 deg
h	0.5 TO 1.0 inch
ϕ	0.5 TO 1.0

Figure 3. Subsonic Flameholder Test Configuration

85-6-51-3

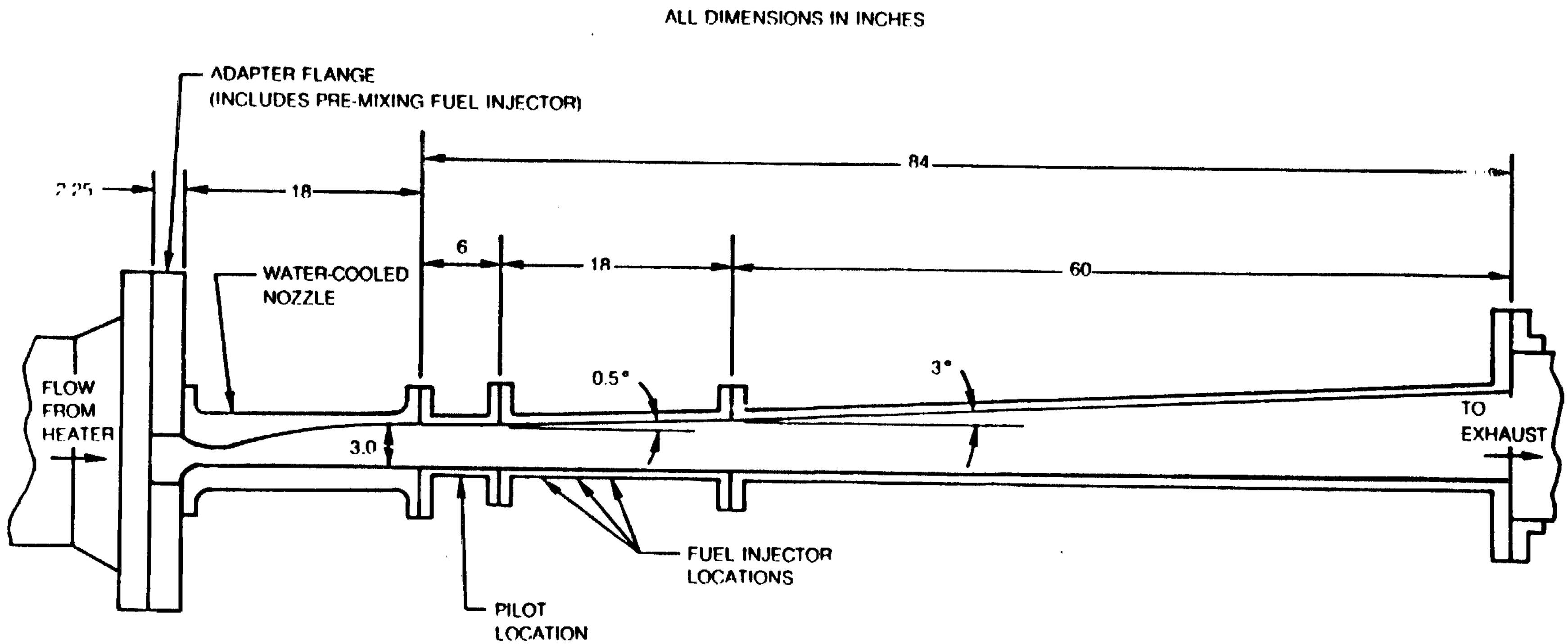


Figure 4. Supersonic Combustion Research Model Geometry

85–6–51–1

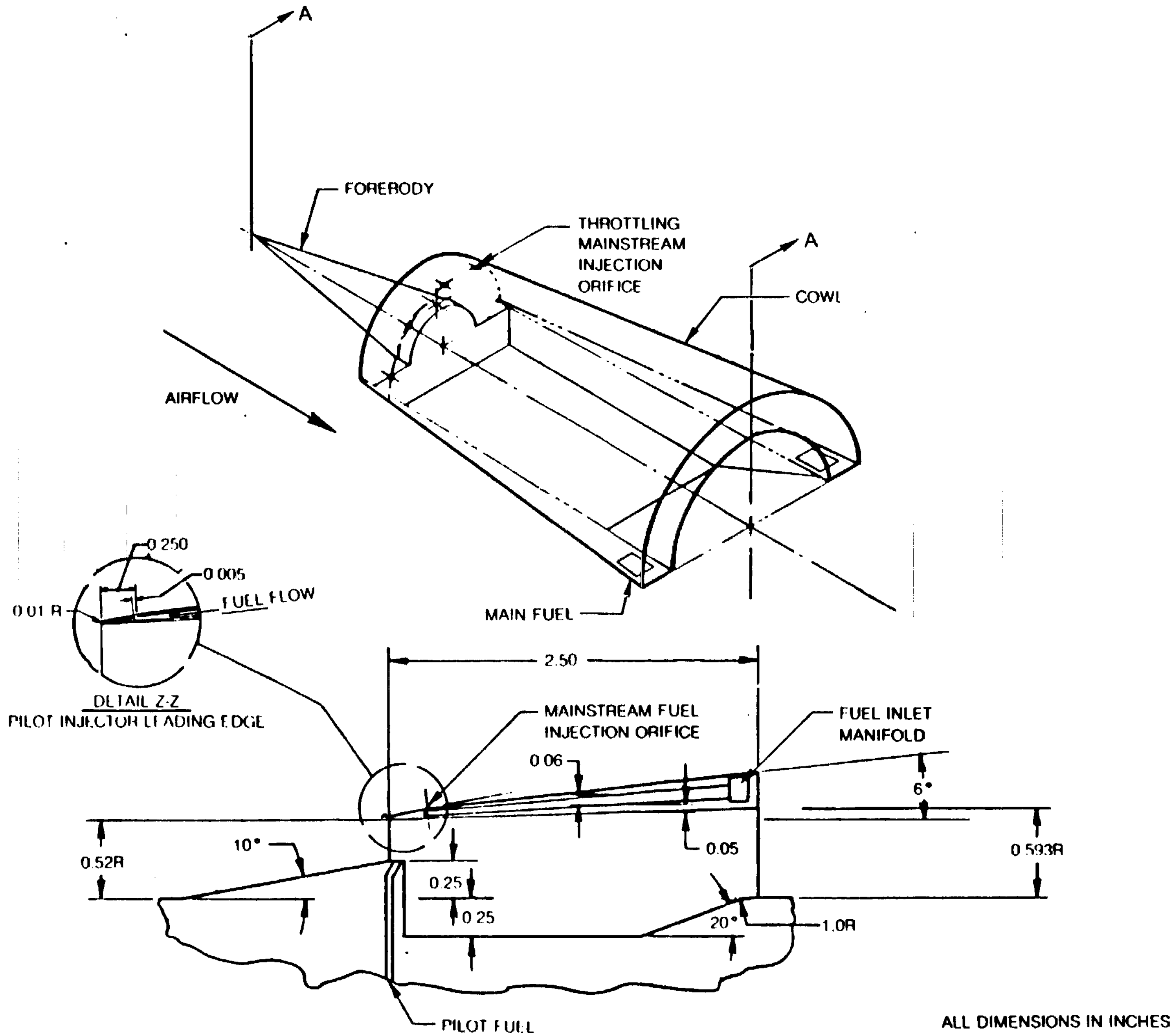


Figure 5. Flash — Vaporizing Piloting Fuel Injector Configuration

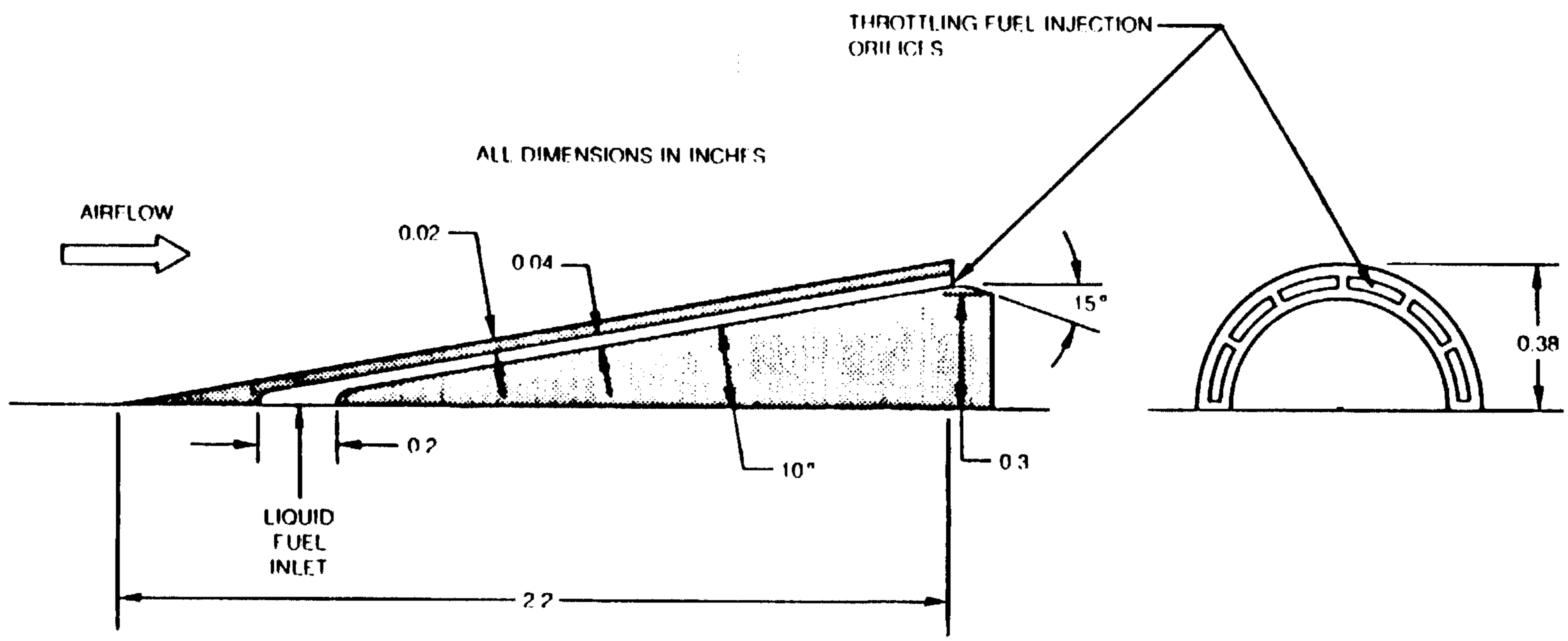


Figure 6. Flash — Vaporizing Fuel Injector Configuration

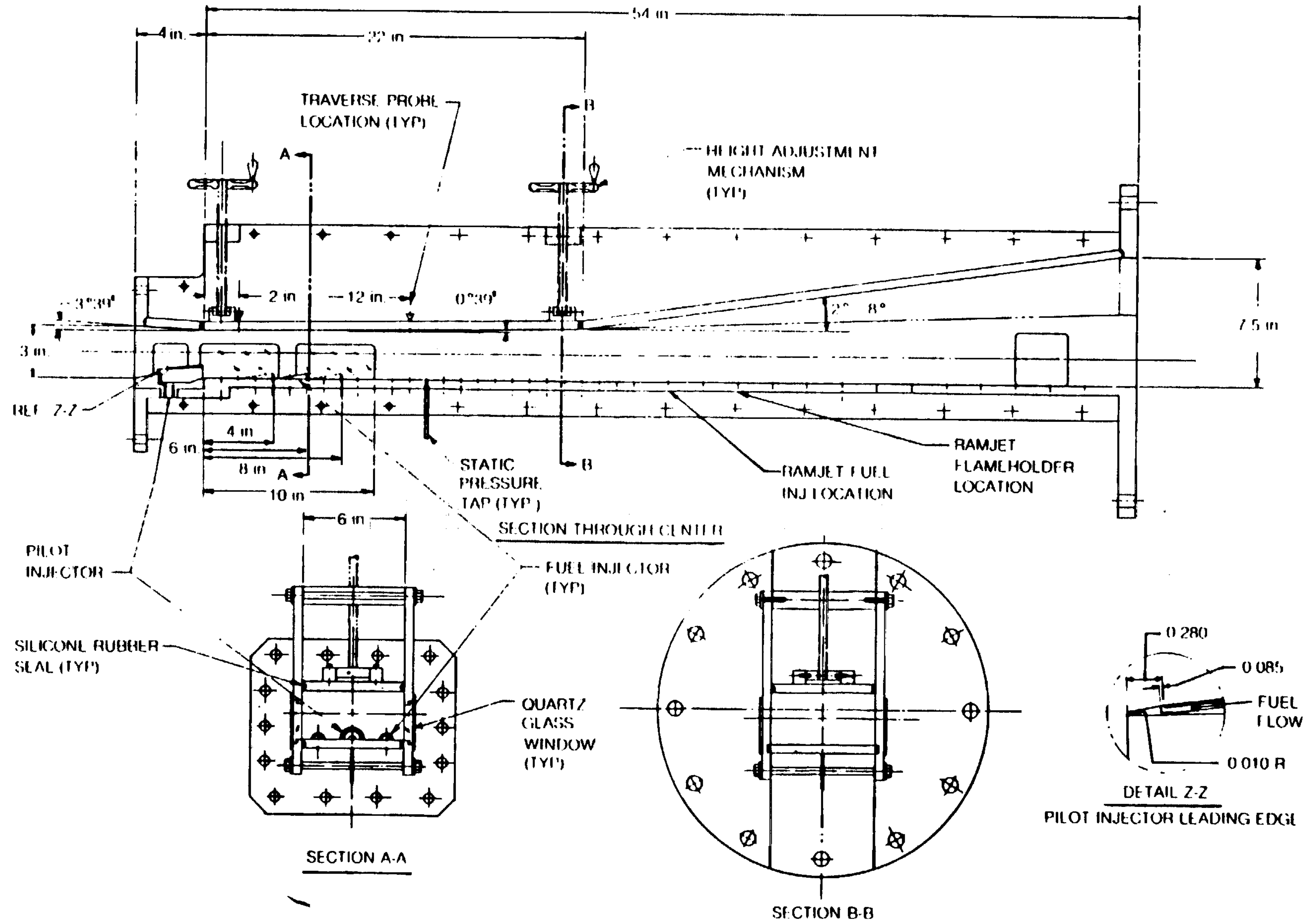


Figure 7. Variable — Geometry Two-Dimensional Combustor Model

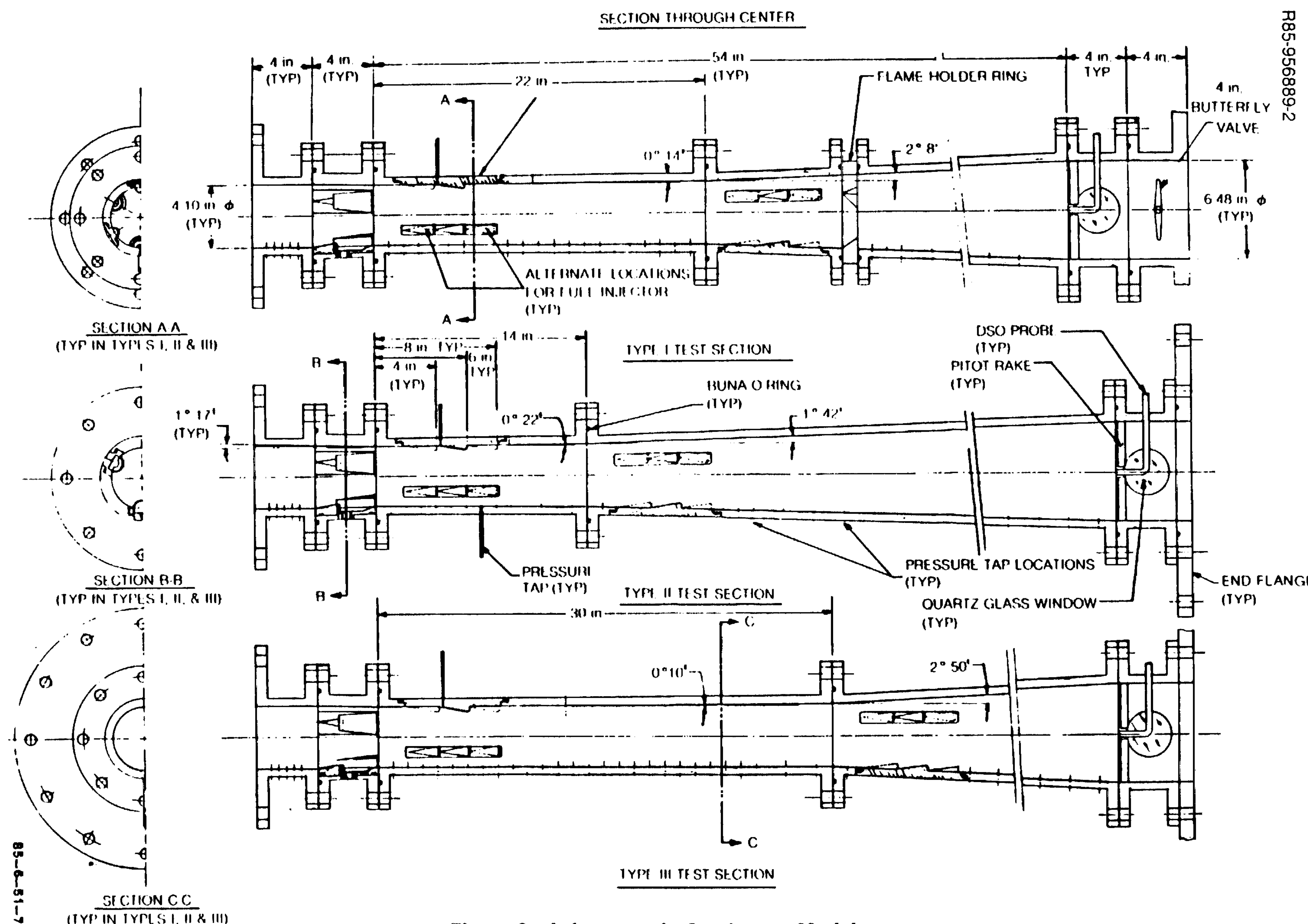


Figure 8. Axisymmetric Combustor Model

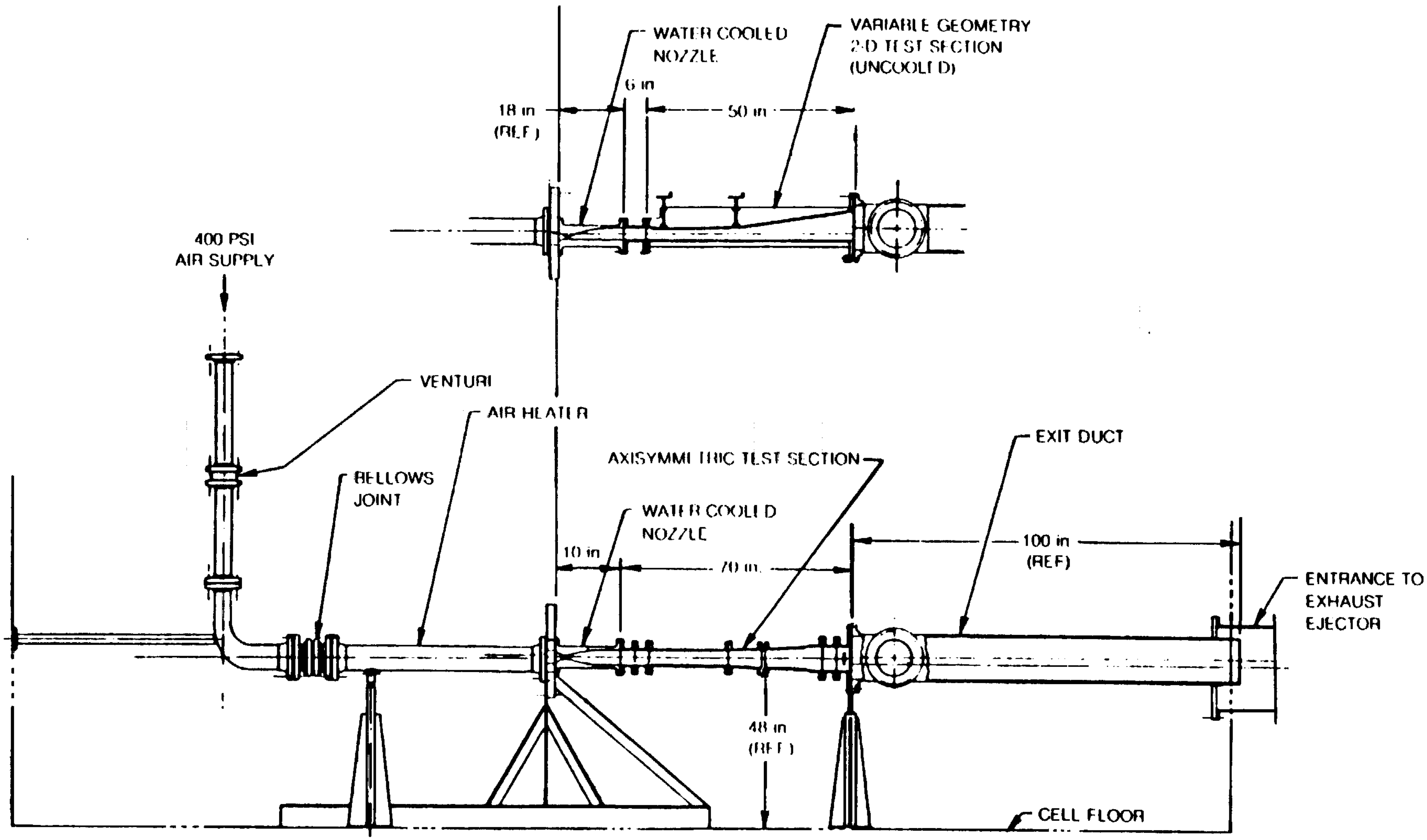
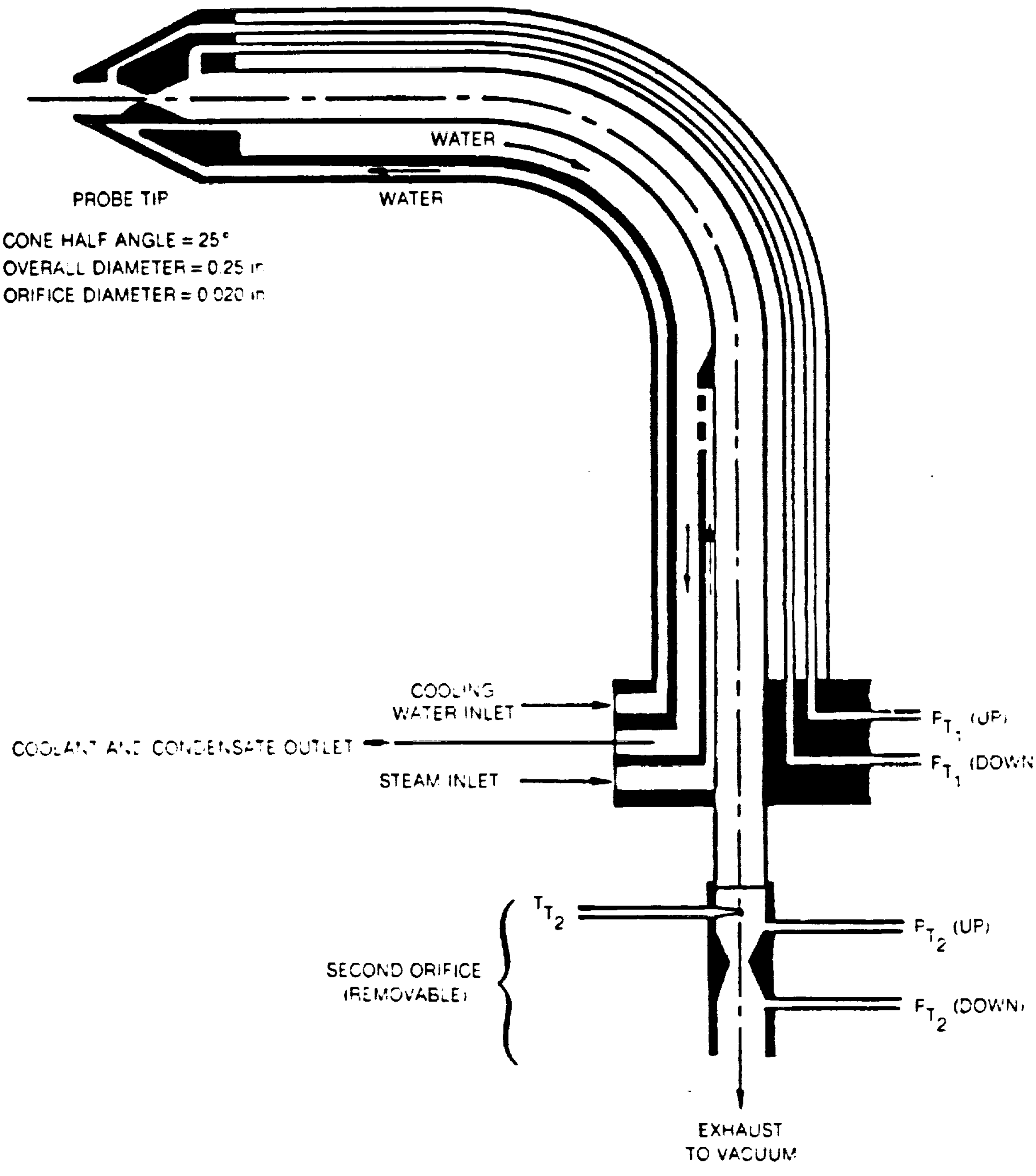


Figure 9. Test Facility Installation

85–6–51–8



NOTE: SUBSCRIPTS SHOWN ON THIS FIGURE REFER TO 1st AND 2nd ORIFICES

Figure 10. Double Sonic Orifice Probe

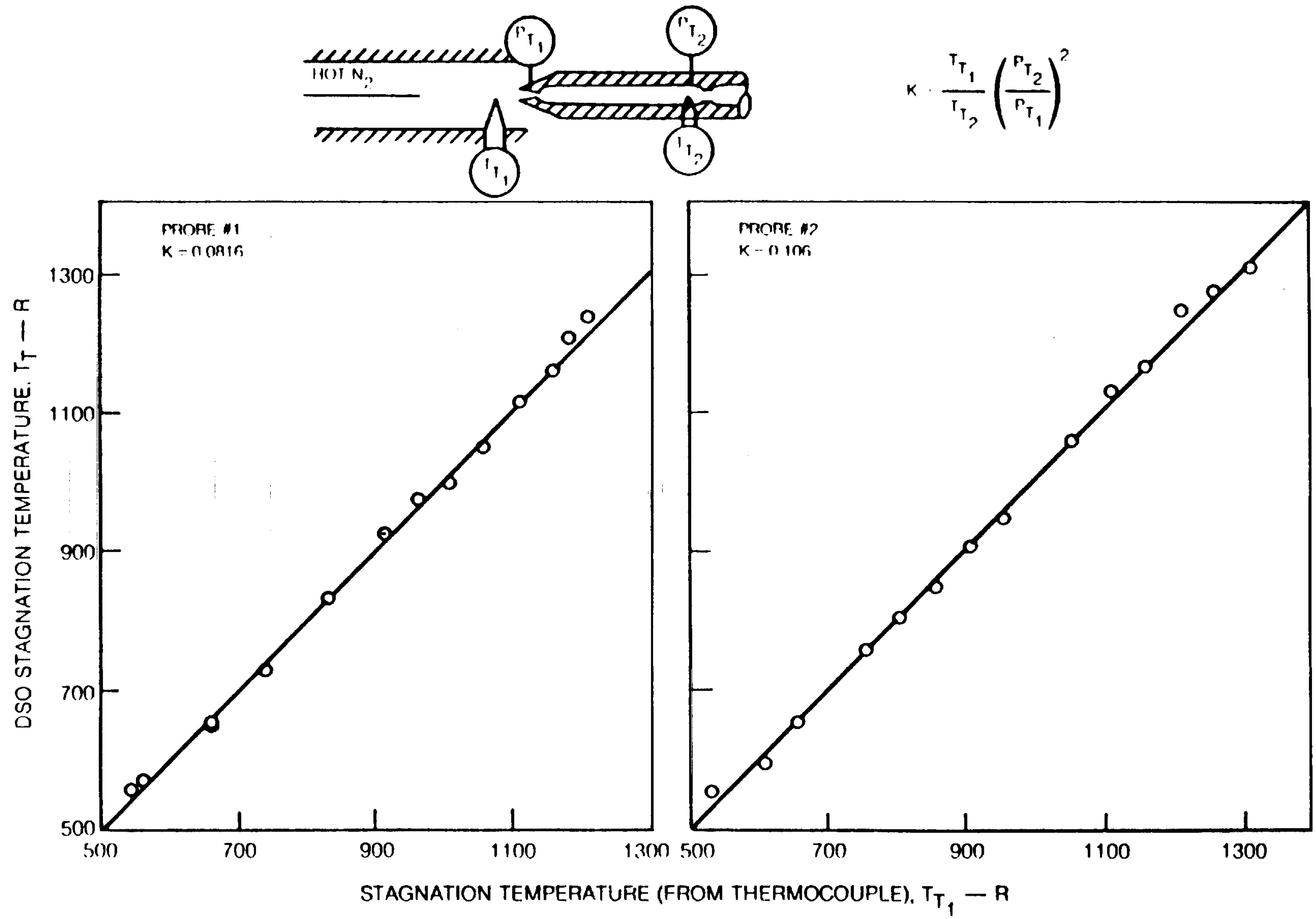


Figure 11. DSO Probe Calibration

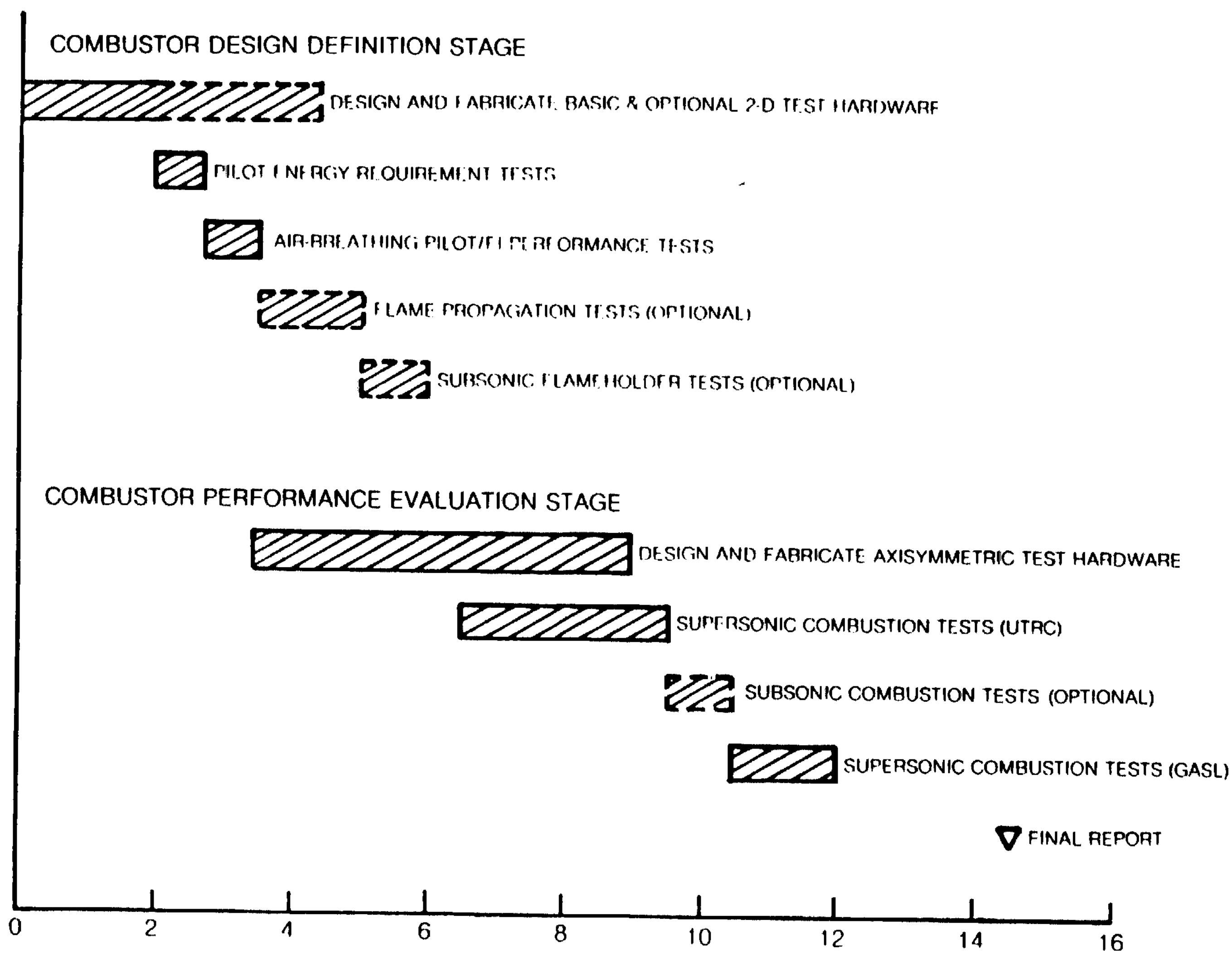


Figure 12. Phase II Test Schedule